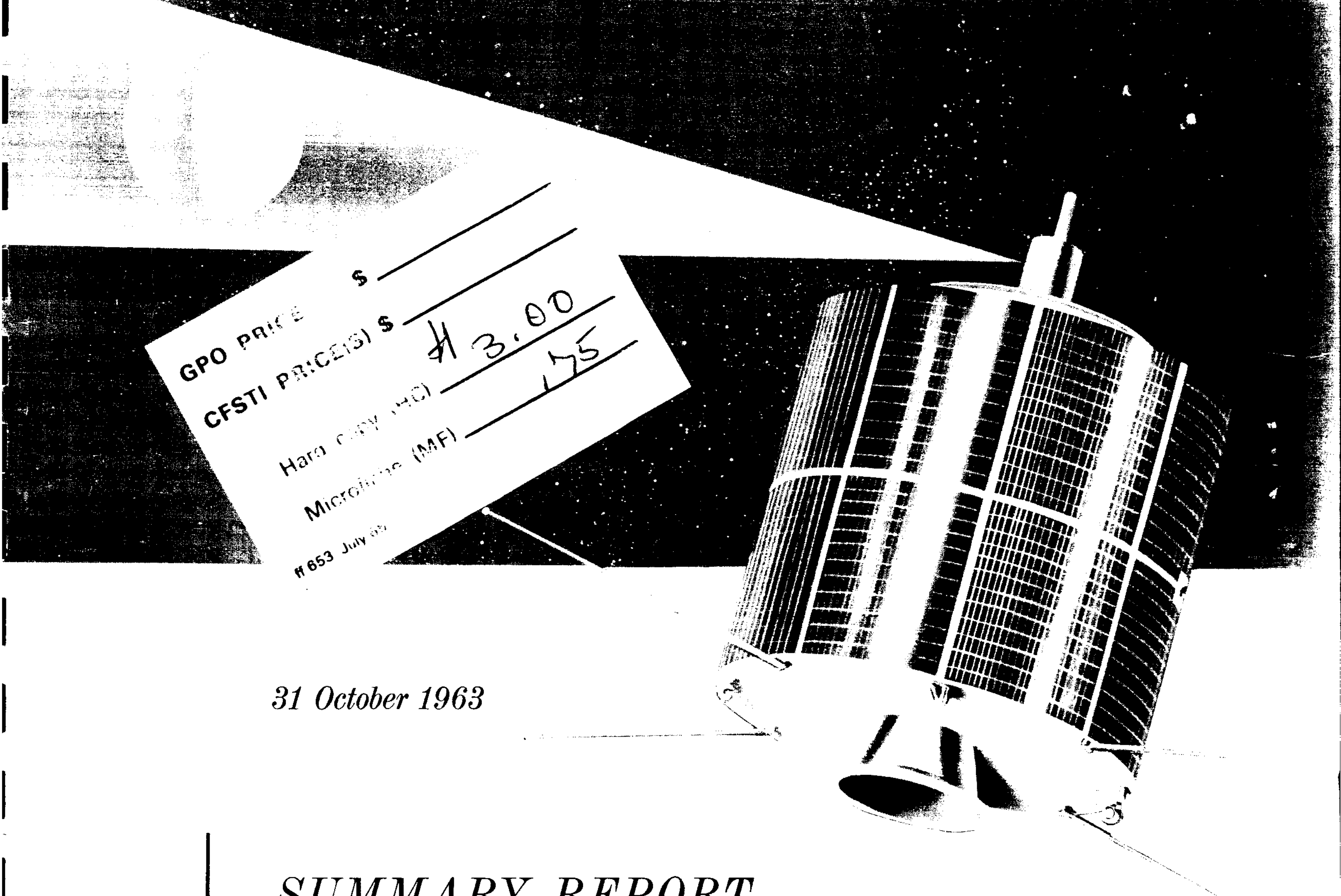
Advanced SYNCOM



31 October 1963

SUMMARY REPORT

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Volume 4

•

System and Subsystem Performance Requirements

•

NASA Contract 5-2797

HUGHES

HUGHES AIRCRAFT COMPANY
SPACE SYSTEMS DIVISION

SSD 31121R

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SUMMARY REPORT

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1. INTRODUCTION

The use of communication satellites provides the practical solution to the need for greatly expanded global communications capability. A major effort of the United States Government and of industry has been in process since the late 1950s to develop a satellite relay system at the earliest possible time.

The National Aeronautics and Space Administration, having management responsibility for developing the space technology leading to a communication satellite system, has investigated nonsynchronous passive satellites and nonsynchronous active repeater satellites. The Goddard Space Flight Center Project Syncom is assigned the synchronous-orbit active repeater satellite investigations.

Under NASA Goddard Space Flight Center Contract NAS5-1560, Hughes developed and constructed three Syncom spacecraft for launch from the Atlantic Missile Range by Delta Launch vehicles for conduct of inclined synchronous-orbit communications experiments during 1963. The Syncom spacecraft have demonstrated a simple spin-stabilized, active repeater satellite design capable of being placed in a synchronous orbit. Similarly, it has been demonstrated that a simple pulsejet control system can provide the stationkeeping necessary to maintain a synchronous orbit.

The Advanced Syncom spacecraft, currently under study for feasibility and advanced technological development, will demonstrate the stationary, or equatorial, synchronous orbit with a vehicle providing a relatively large, adaptable payload capability, achievement of long life in orbit, an electronically steerable antenna beam, continuous wide-band communications, and new multiple-access communications. Scientific instruments will be carried to measure the radiation environment and to assess radiation damage occurring during the orbiting process and throughout satellite life in the synchronous, equatorial orbit.

The Advanced Syncom study program has included research and development of engineering models of a multielement phased array transmitting antenna and associated control circuits; a dual-mode communications transponder operating at 6-gc receiving frequency and 4-gc transmitting frequency and providing alternate modes of operation as a multi-channel SSB-PM multiple-access transponder or as a wide-band FM frequency translation transponder; a traveling-wave tube final power amplifier for the transponders; a spacecraft structure; and a bipropellant rocket jet control system.

In May 1963, the NASA expanded the advanced technology program to include design effort on all elements of the spacecraft and the communication system test equipment. Currently under way is the fabrication and testing of advanced engineering models of the communication transponders, transmitting and receiving antennas, and traveling-wave tube power amplifiers. Breadboard circuits of the telemetry encoders and command decoders are similarly in process. The system test equipment being developed will permit quantitative measurements of communication system performance to be obtained. The above activities will be completed by the end of October 1963.

This Summary Report covers the technical progress achieved during the contract period and details the system configuration and specifications resulting from system studies. The report is divided into seven volumes:

Volume 1: Advanced Syncom Summary Report

Volume 2: Major and Minor Control Item Test Plans and In-process Specifications

Volume 3: Interface Reports

Volume 4: System and Subsystem Performance Requirements

Volume 5: System Test Plans

Volume 6: Engineering Data on One Set of Transponder Control Items

Volume 7: T-1 Structural Vibration Test Report

2. SPACECRAFT SYSTEM SUMMARY

The Advanced Syncom satellites, as in the Delta-launched Syncom, will utilize spin stabilization for attitude stabilization. The spacecraft physical parameters are increased over those of Syncom to accommodate a large increase in communications capacity. In addition, the self-contained apogee injection stage removes approximately 29 degrees of inclination from the orbit while circularizing the elliptical transfer orbit at the synchronous radius. The parameters of an Advanced Syncom spacecraft are summarized in Table 2-1. Figures 2-1 and 2-2 illustrate the general arrangement, and the structural engineering model of the Advanced Syncom.

The communication capacity of each of the satellite transponders is 600 two-way telephone conversations. The design for each satellite contains four such transponders, providing a total system capacity through the satellite of 2400 two-way voice channels. Alternately, the system can accommodate television or other wide-bandwidth signals through each of the transponders.

Ground-station characteristics for which the full communications capacity is achieved would be as follows:

1) Transmitter (for each frequency assignment): saturated power, 10 kilowatts; frequency band, 6 gc; bandwidth, 25 mc; diplexer loss, -1 db; and frequency stability, 1 part in 10^{10} for short term and 1 part in 10^7 for long term.

2) Antenna: diameter, 85 feet; efficiency (transmitting and receiving), 54 percent.

3) Receiver noise temperature (all sources including antenna), 80°k.

Smaller stations can be used with a proportionate reduction in capacity. With 40-foot-diameter antennas and the same transmitters and receivers, the voice channel capacity is reduced to 120 two-way channels and the television signal noise level, and therefore picture quality, falls below CCIR standards.

The system has two alternate modes of operation possible in each assigned frequency band. The first mode accommodates a wide-band FM transmission to the spacecraft, which translates the signal-carrier frequency and repeats the signal with no conversion in modulation. This mode is used for television or other wide-band data originating from a single station. The spacecraft transponder mode for such signals is termed the "frequency-translation mode."

The second mode of operation involves the transmission, simultaneously from a large number of ground stations, of frequency division multiplexed, single-side-band, suppressed carrier voice channels. The signals are converted into phase modulation of a single carrier in the spacecraft, and are retransmitted back to all stations in this form. This mode permits simultaneous two-way interconnection of all combinations of the ground stations. The spacecraft transponder mode for these signals is termed the multiple-access mode.

TABLE 2-1. PARAMETERS OF ADVANCED SYNCOM

Physical Configuration	58-inch diameter cylinder
Weight	1500 pounds at launch
	760 pounds in 24-hour, equatorial orbit
Apogee injection motor	Solid propellant, JPL
Control systems	Liquid bipropellant
	Fuel: monomethylhydrazine
	Oxidizer: nitrogen-tetroxide
	Two independent systems: capacity per system adequate for correcting initial errors and providing 3 years stationkeeping
	Self-contained spin rate control
Communications	Four independent dual-mode transponders
	Redundant 4.0-watt traveling-wave tube power amplifiers in each transponder
	8 db collinear array receiving antenna
	18-db phased array transmitting antenna
	6.02 to 6.3 gc ground-to-spacecraft
	3.99 to 4.18 gc spacecraft-to-ground
Telemetry	Four 1.25-watt transmitters in 136-mc band
	Four encoders; GSFC PFM standard
Command	Four receivers in 148-mc band
	Four decoders; GSFC FSK standard
Electrical power	147-watt, N-P solar cell array
	650-watt-hour rechargeable nickel-cadmium energy storage system

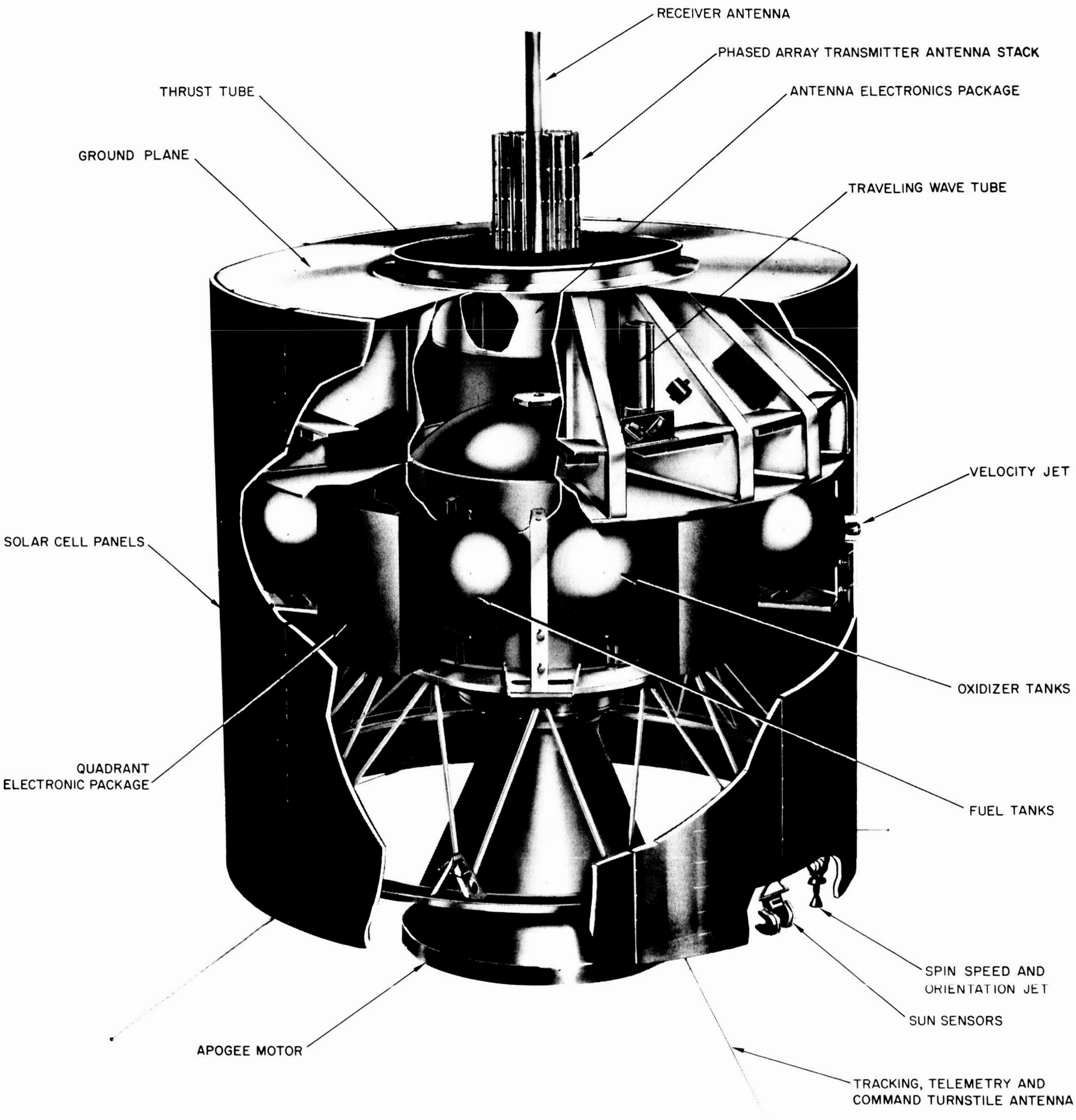

Figure 2-1. General Internal Arrangement of Advanced Syncom

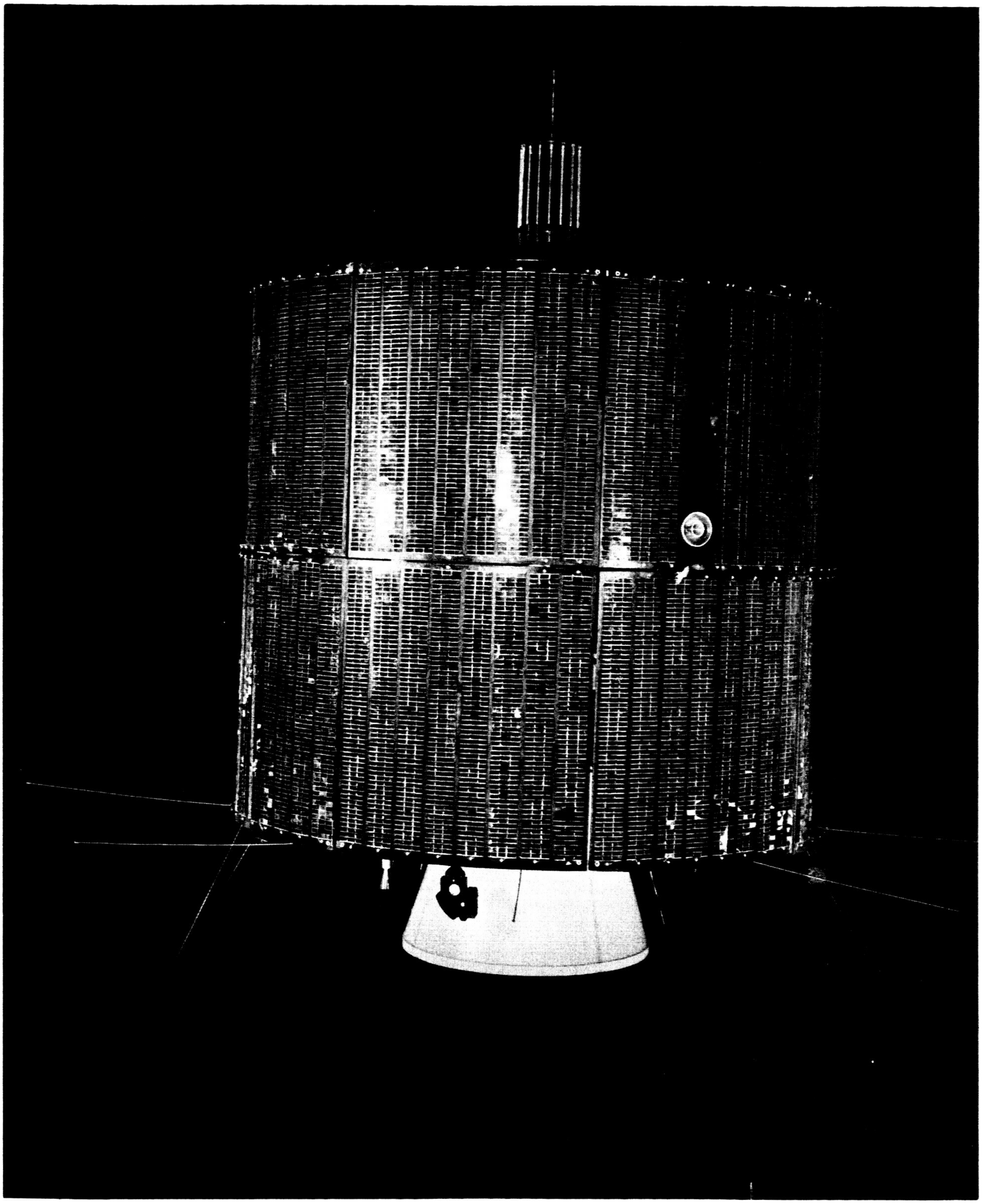

Figure 2-2. Structural Engineering Model of Advanced Syncom

ADVANCED SYNCOM
SYSTEM PERFORMANCE SPECIFICATION
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3. ADVANCED SYNCOM SYSTEM PERFORMANCE SPECIFICATION S2-100

1. SCOPE

This specification establishes the system requirements for the Advanced Syncom spacecraft. The Advanced Syncom is an active communications spacecraft project of the National Aeronautics and Space Administration, Goddard Space Flight Center.

2. APPLICABLE DOCUMENTS

The following documents form a part of this specification to the extent specified herein:

2.1 NASA/GSFC Field Projects Branch, Cape Canaveral

Dated 16 April 1963
Spacecraft Operations at AMR,
Safety Regulations and Documentation Requirements

2.2 Hughes Aircraft Company Interface Reports

Dated 1963
Atlas-Agena D/Spacecraft Interface Report

Dated 1963
Spacecraft/Apogee Injection Rocket Motor Interface Report

2.3 Spacecraft Subsystems Performance Requirements Specification, Current Revision

2.4 General Arrangement Drawing

Spacecraft Designation	Applicable Drawing
T-2	Hughes Aircraft Company X209811, current revision

Spacecraft Designation	Applicable Drawing
T-3	Hughes Aircraft Company X209811, current revision
T-4	
Y-1	
F-1	
F-2	
F-3	
F-4	

2.5 Orientation Drawing

Spacecraft Designation	Applicable Drawing
T-2	Hughes Aircraft Company X202865, current revision
T-3	
T-4	
Y-1	
F-1	
F-2	
F-3	
F-4	

3. REQUIREMENTS

3.1 Orbital Requirements

3.1.1 Spin Stabilization. The spacecraft shall be a spin-stabilized vehicle.

3.1.1.1 Spin Maintenance Impulse Requirement: To maintain the nominal synchronous orbit injection spin speed, a total impulse of 1000 pound-seconds shall be provided.

3.1.1.2 Spin Speed Control: The spacecraft shall be capable of proportional (unsaturated) control of the spin speed over the range of 100 ± 25 rpm and shall remain operative over the range 100 ± 50 rpm.

3.1.1.3 Spin Speed Resolution: The spacecraft shall be capable of resolving the spacecraft spin speed to ± 5 rpm within the range of 100 ± 25 rpm.

3.1.2 Orbit Injection

3.1.2.1 Transfer Orbit Injection: The spacecraft shall be capable of being launched into a transfer orbit by an Atlas-Agena D vehicle.

3.1.2.1.1 Spin-Up: The Agena D shall impart a nominal spin of 100 rpm to the spacecraft prior to separation. The spacecraft shall be capable of operation under the spin-up accelerations defined in the Atlas-Agena D/Spacecraft Interface Report.

3.1.2.2 Synchronous, Circular, Equatorial Orbit Injection

3.1.2.2.1 Nominal Orbit Injection: A solid-propellant rocket motor, which can provide a nominal velocity increment of 6100 fps to place the spacecraft in a synchronous, circular orbit in the earth's equatorial plane from an inclined, elliptical transfer orbit, shall be provided to the Hughes Aircraft Company. The maximum weight of the motor shall not exceed 874.5 pounds before firing or 102 pounds after firing. The following parameters, applicable to the detail design of the solid-propellant rocket motor, shall be defined as indicated below:

1) Transfer-ellipse semi-major axis

2) Transfer-ellipse eccentricity

3) Transfer-ellipse inclination

4) Transfer-ellipse spin-axis orientation

   These items are defined in the Atlas-Agena D/Spacecraft Interface Report

5) Maximum spacecraft weight: defined in Advanced Syncom Systems Specification

6) Rocket motor weight (before and after firing): defined in Spacecraft/Apogee Injection Motor Interface Report

The rocket motor shall be capable of being ignited by an on-board fixed timer or by command from the ground.

3.1.2.2.2 Orbital Vernier Corrections: A total velocity increment of 250 fps shall be provided to correct 3σ ascent and apogee injection motor guidance errors defined as stated below:

Ascent guidance errors - Atlas-Agena D/Spacecraft Interface Report

Apogee motor guidance error - Spacecraft/Apogee Injection Motor Interface Report

These errors are nominally a longitudinal drift rate of 14.65 degrees per day; orbital inclination of 0.63 degrees; orbital eccentricity of 0.0223; spin axis attitude error of 1 degree.

3.1.3 Orbit Maintenance (Stationkeeping). To keep the spacecraft fixed within 0.05 degree of a subsatellite point, a total velocity increment of 935 fps shall be provided to correct sun-moon and triaxiality perturbations.

3.1.4 Spacecraft Spin-Axis Orientation. A total impulse of 427 pound-seconds shall be provided to initially orient the spacecraft spin axis normal to the orbital plane.

A total impulse of 95 pound-seconds shall be provided to maintain spin-axis orientation so as to permit on-board equipment to direct a communications beam onto a desired line-of-sight area of the earth's surface.

3.1.5 Spacecraft Orbital Control. Orbital control equipment shall be provided to perform orbital vernier corrections, orbit maintenance (stationkeeping) and spin-axis orientation as required.

3.2 Mechanical Requirements

3.2.1 Dimensions

3.2.1.1 Static Envelope: The static envelope of the spacecraft shall be within the size limitations defined by the current revision of Hughes Aircraft Drawing No. X209783.

3.2.1.2 Dynamic Envelope: The dynamic envelope of the spacecraft shall be within the size limitations defined by the current revision of Hughes Aircraft Drawing No. P207103.

3.2.2 Weight. The weight of the spacecraft shall be 1518 pounds including the apogee injection motor and ballast. The weight of the apogee injection motor shall be defined in the Spacecraft/Apogee Injection Rocket Motor Interface Report but shall in no case exceed 874.5 pounds when fully loaded nor 102 pounds after burnout.

3.2.3 Balance

3.2.3.1 Static Balance: The center of gravity offset measured normal to the spacecraft longitudinal geometric axis shall not exceed the appropriate value shown in Table 1.

3.2.3.2 Dynamic Balance: The tilt of the spacecraft spin axis with respect to the spacecraft longitudinal geometric axis shall not exceed the appropriate value shown in Table 1.

3.2.4 Spin Stability

3.2.4.1 Ratio of Roll Moment of Inertia to Pitch Moment of Inertia: The ratio of the spacecraft roll moment of inertia (about the spin axis) to the largest moment of inertia about any axis lying in a plane normal to the spin axis and passing through the center of gravity shall be not less than the appropriate value shown in Table 1. The balance and spin stability requirements of this table apply only when the apogee injection motor meets the balance requirements defined in the Spacecraft/Apogee Injection Rocket Motor Interface Report and in no case exceeds the values shown in Table 2.

TABLE 2. APOGEE INJECTION MOTOR BALANCE REQUIREMENTS

Apogee Injection Motor Mass Properties	Values
Static balance	
Dynamic balance	
Minimum ratio $\frac{\text{roll moment of inertia}}{\text{pitch moment of inertia}}$	

3.2.4.2 Nutation

3.2.4.2.1 Nutation Angle: The nutation of the spacecraft shall be constrained so as not to exceed the following operational requirements:

1) The loss in apogee injection rocket motor thrust due to nutation shall not exceed 900 pound seconds.

2) The nutation of the spacecraft when on station shall not cause the amplitude modulation of signals, received at two earth stations located at opposite ends of the continuous visibility cone formed by the spacecraft and the locus of 5 degrees minimum station elevation lines, to exceed 2 db.

TABLE I. SPACECRAFT BALANCE REQUIREMENTS

Spacecraft Configuration / Parameter Specified	Prior to Installation of Apogee Motor and Fueling of Bipropellant Reaction Control System	After Separation But Before Apogee Injection Motor Ignition	During Apogee Injection Motor Burn	After Apogee Motor Burnout	50 Percent Depletion of Reaction Control System Propellants	Complete Depletion of Reaction Control System Propellants
Static balance - maximum center of gravity offset normal to longitudinal geometric axis, inches	0.005	0.043	0.043	0.036	0.033	0.031
Dynamic balance - maximum spin axis tilt, milliradians	2.0	3.5	5	4.5	4.5	4.5
Minimum ratio, roll moment of inertia to pitch moment of inertia.		1.15		1.20	1.15	1.10

3.2.4.2.2 Nutation Time Constant: The nutation time constant shall be less than 1 hour. This will allow the spacecraft to meet the orbital requirements of paragraph 3.1.

3.2.5 Configuration

3.2.5.1 Spacecraft Subsystems: The spacecraft shall consist of the following subsystems:

1) Spacecraft structure subsystem

2) Antenna control subsystem

3) Telemetry and command subsystem

4) Power supply subsystem

5) Communication subsystem

6) Wire harness subsystem

7) Reaction control subsystem

8) Apogee injection motor subsystem

The operation of these subsystems shall be as specified in the current revision of the Subsystem Performance Requirements Specification.

3.2.5.2 Structure: The spacecraft structure shall provide basic support for the other subsystems of the spacecraft and for attachment to the spacecraft support structure of the boost vehicle.

3.2.5.2.1 Appendages: Telemetry-command antenna whips, axial jets, sun sensors, the battery-charging umbilical and disconnect plug, the squibs shorting plug, and the test access receptacles shall be mounted on the forward (+Z direction) end of the spacecraft. The communications antennas and separation switch actuators shall be mounted on the aft (-Z directions) end of the spacecraft. Radial jets shall protrude from the periphery of the spacecraft.

3.2.5.3 Packaging

3.2.5.3.1 Interchangeability: The spacecraft major control items shall be designed to permit their replacement during prelaunch tests with a minimum adjustment of other major control items.

3.2.5.3.2 Mounting: All major control items shall be hard mounted.

3.2.5.4 Orientation: The orientation of all items of the spacecraft shall be as defined in the current issue of Orientation Drawing Hughes Aircraft Company 202865.

3.2.6 Thermal Control Requirements. Thermal control of the spacecraft shall be accomplished by passive and/or active means. Temperatures shall be controlled on each spacecraft control item or component part, to within a range compatible with its function and reliability requirements, while operating in the temperature environment defined in this specification.

3.2.7 Safety

3.2.7.1 Pressurization: The spacecraft subsystems shall be designed to operate without pressurization. The bipropellant reaction control subsystem and battery cells shall be exempted from this requirement but the bipropellant reaction control subsystem shall meet the requirements for pressure systems as defined in "Spacecraft Operation at AMR".

3.2.7.2 Ordnance Devices: All ordnance devices of the spacecraft must satisfy the requirements for ordnance devices called out in "Spacecraft Operations at AMR".

3.3 Electrical Requirements

3.3.1 General

3.3.1.1 Spin-Speed Operation: All control items shall be capable of operation within specification while being subjected to "g" fields resulting from spin speeds and spin accelerations called out in this specification.

3.3.1.2 Redundancy: Redundancy shall be employed in the spacecraft design whenever a significant increase in survival probability can be obtained without violating any of the other provisions of this specification.

3.3.1.3 Electromagnetic Interference: During normal operation there shall be no impairment beyond specified levels of any spacecraft function due to electromagnetic interference.

3.3.2 Communication Requirements. Facilities to receive, convert frequency, amplify, and retransmit microwave signals shall be provided. Four transponders shall be provided. Each transponder shall be capable of operating in either an FM-to-FM wideband mode (frequency translation) or a SSB, frequency-multiplex-to-phase-modulated frequency-multiplex mode (multiple access). In the frequency translation mode, the spacecraft shall be capable of transmitting an unmodulated RF signal (beacon). The frequency assignments of each transponder shall be as defined in the current revision of the Spacecraft Subsystems Performance Requirements Specification.

3.3.3 Command Requirements. RF receiver and decoder groups shall be provided for accepting and executing commands to control spacecraft functions from the ground. The command format shall be as defined in the Spacecraft Subsystems Performance Requirements Specification.

3.3.4 Telemetry Requirements. Encoders and RF transmitters shall be provided to encode and transmit to ground stations spacecraft measurements and control data necessary to assess spacecraft performance. The on/off transmission of this data shall be controlled by ground command. The telemetry data format shall be as defined in the Spacecraft Subsystems Performance Requirements Specification.

3.3.5 Electrical Power. Solar cells shall be provided as a primary source of electrical energy to power the spacecraft subsystems. Battery cells shall be provided for electrical power during transient loads and solar eclipse.

3.3.5.1 Sun Angle: The solar panel power supply shall perform within requirements defined in the Subsystem Performance Requirements Specification for any angle between the spacecraft spin axis and the sun line of 90 ± 25 degrees.

3.3.5.2 Battery Charging: Sufficient solar cell capacity shall be provided to initially allow for battery charging without interruption of any spacecraft function. Additional capacity to compensate for solar-cell degradation shall be provided to a limit not to exceed weight, static and dynamic envelope, and all other pertinent requirements of this specification.

3.3.6 Spacecraft Modes of Operation

3.3.6.1 Communication

3.3.6.1.1 Transponder: The communications transponders shall be capable of continuous operation, singly or in combination, in either a frequency translation or multiple access mode.

3.3.6.1.2 Beam Transmission: The spacecraft shall be capable of transmitting a pancake or directional communications RF beam.

3.3.6.2 Command

3.3.6.2.1 Command Receivers: The command receivers shall operate continuously.

3.3.6.2.2 Command Decoders: A command decoder when addressed shall operate in a shift register mode or a count mode.

3.3.6.3 Telemetry: The telemetry transmitter/encoder groups shall be capable of continuous operation singly or in pairs.

3.4 Environmental Requirements

3.4.1 Prelaunch Environment

3.4.1.1 General: The spacecraft shall be kept in an air conditioned environment at all times until lift-off. During transit it shall be kept covered to give it protection from the elements.

3.4.1.2 Thermal Environment

3.4.1.2.1 General - Propellant Tanks Loaded: When the bipropellant tanks are filled with fuel and oxidizer the spacecraft must operate within specifications over the temperature range of 40° to 80°F.

3.4.1.2.2 General - Bipropellant Tanks Unloaded: When the bipropellant tanks are not filled with fuel and oxidizer the spacecraft must operate within specification over the temperature range of 50° to 100°F.

3.4.1.3 Humidity: When not in operation the spacecraft must survive without damage relative humidity up to 100 percent at temperatures such that condensation will take place in the form of water. The spacecraft shall be capable of operation in an environment having a relative humidity of 50 percent.

3.4.1.4 Vibration: All movement of the spacecraft shall be controlled either by choice of transportation and/or by use of suitable fixtures such that the vibration input shall be less than the predicted vibration input during launch.

3.4.2 Launch Environment (Lift-Off to Separation). The spacecraft must be capable of operation while being subjected to the environment specified below.

3.4.2.1 Thermal Environment: The spacecraft shall be capable of operation during launch when the maximum temperature exposure of the outer surfaces of the spacecraft is expected to be 165°F, based on a maximum fairing temperature of 300°F, a fairing jettison time of approximately 350 seconds from lift-off, and a nominal jettison altitude of 518,000 feet.

3.4.2.2 Ambient Pressure Environment: The spacecraft shall be capable of operation while operating in an ambient pressure environment which varies from sea level to 1×10^{-10} Torr.

3.4.2.3 Vibration: The Goddard Space Flight Center has provided the following qualification vibration levels for evaluation of the Advanced Syncom structure. Levels to be applied at the separation interface include sinusoidal excitation, three axes, logarithmic sweep at two octaves per minute, 4.35 minutes duration.

5 to 15	cps	0.25 inch double amplitude
15 to 250	cps	3 g peak
250 to 400	cps	5 g peak
400 to 2000	cps	7.5 g peak

Random excitation, three axes, 6 minutes duration per axis.

20 to 80	cps	$0.04\ g^2/cps$
80 to 1280	cps	Increasing from $0.04\ g^2/cps$ at 0.61 db per octave
1280 to 2000	cps	$0.07\ g^2/cps$

Exception to the above may be taken for both sinusoidal and random excitation where the predominant longitudinal and lateral frequencies are sufficiently decoupled from those of the Atlas-Agena with the spacecraft attached. In this case, the spacecraft response at the center of gravity in the bandwidth of the predominant frequency may be limited to that of the separation plane input in the same predominant frequency bandwidth.

Torsional vibration, applied at the separation plane and corrected to the interface diameter assuming rigid body motion, sinusoidal excitation, logarithmic sweep at one octave per minute.

50 to 80 cps	7.5 g peak tangential or 5-foot diameter

3.4.2.4 Acceleration

3.4.2.4.1 Load Factors: The spacecraft shall be capable of operation while experiencing the following values of axial and side load factors due to sustained accelerations:

3.5 g along the transverse axes

+7.5 g along the longitudinal axis

3.4.2.4.2 Angular Acceleration: The spacecraft shall be capable of operation while experiencing an angular acceleration of not greater than 33 rad/sec^2.

3.4.3 Orbital Environment (after Separation)

3.4.3.1 Thermal Environment

3.4.3.1.1 Thermal Environment (Transfer Ellipse): The spacecraft shall be capable of operation while in the thermal environment of the transfer ellipse.

3.4.3.1.2 Thermal Environment (Stationary Orbit): The spacecraft shall be capable of operation within specification when in a thermal environment in which the external surfaces of the spacecraft are subject to radiation from the sun (nominally 442 Btu/ft^2/hr) radiation from the earth (including reflected sunlight) and radiation to space. The spacecraft shall also be capable of operation within specification during solar eclipses to be expected when in a stationary, synchronous orbit.

3.4.3.2 Ambient Pressure Environment: The spacecraft shall be capable of operation within specification at the synchronous altitude where the ambient pressure is expected to be 1×10^{-10} Torr.

3.4.3.3 Longitudinal Acceleration: The spacecraft shall be capable of operation while experiencing a longitudinal acceleration, in the -Z direction due to firing of the apogee motor, of not greater than 8.5 g.

3.4.3.4 Radiation Environment

3.4.3.4.1 General: The spacecraft shall be designed such that the most detrimental radiation environment described in the following paragraphs shall not impair spacecraft functions beyond specification levels.

3.4.3.4.2 Transfer Ellipse: The proton and electron particle energies and integrated flux to be encountered during the parking orbit and transfer ellipse are estimated to be:

Particle	Particle Energy, electron volts	Integrated Flux, particles/cm^2
Electrons	$\geq 40 \times 10^3$	1×10^6
	$\geq 40 \times 10^6$	6×10^{12}
Protons	$(0.1 \text{ to } 5) \times 10^6$	3×10^{12}
	$\geq 30 \times 10^6$	6×10^5

3.4.3.4.3 Synchronous Orbit Radiation Environment

3.4.3.4.3.1 Radiation Due to Outer Van Allen Belt: The proton and electron particle energies and integrated flux to be encountered during a 3-year period in the synchronous orbit due to the outer Van Allen belt is estimated to be:

Particle	Particle Energy, electron volts	3-Year Integrated Flux, particles/cm^2
Electrons	$\geq 40 \times 10^3$	3×10^{10}
	$\geq 40 \times 10^6$	3×10^{15}
Protons	0.1×10^6 to 5×10^6	3×10^{15}
	$\geq 30 \times 10^6$	6×10^8

3.4.3.4.3.2 Radiation Due to Solar Flares: It is expected that during the 3-year period in orbit solar flares of 3^+ magnitude, yielding an integrated flux of 1×10^8 particles/cm^2/event, will occur. Over this 3-year period, the total flux of protons having energies greater than 20×10^6 electron volts is expected to be 6×10^{12} protons/cm^2.

3.4.3.4.3.3 Galactic Cosmic-Ray Radiation: It is estimated that 6×10^8 particles/cm^2, consisting of 79 percent protons, 20 percent alpha particles, and less than 1 percent heavier nuclei will infringe upon the spacecraft during 3 years in stationary orbit.

3.5 Life

The spacecraft shall be designed for a storage life of 1 year and an operational life of 3 years in stationary orbit.

4. PREPARATION FOR DELIVERY

Protection of the spacecraft against harmful environment shall be provided by a shipping and storage container.

5. NOTES

5.1 Definitions

5.1.1 Subsystem. A subsystem is defined as one or more control items connected or associated together to perform an operational function. (Examples: communication subsystem, structure.)

5.1.2 Longitudinal Geometric Axis. The longitudinal geometric axis is defined as the line perpendicular to the separation plane and concentric with the interface diameter as defined by the runout ring.

5.1.3 Nutation Angle. The nutation angle is defined as the angle between instantaneous spin axis and the direction of the angular momentum vector.

5.1.4 Time Constant. A time constant is the time required for a function to reach 63 percent of its final value.

ADVANCED SYNCOM
SUBSYSTEM PERFORMANCE SPECIFICATION
S2-101

CONTENTS

4. ADVANCED SYNCOM SUBSYSTEM PERFORMANCE SPECIFICATION S2-101

1. SCOPE

This specification defines design requirements for each subsystem of the Advanced Syncom.

2. APPLICABLE DOCUMENTS

2.1 The governing document of this specification shall be:

S2-100	Advanced Syncom System Performance Specification, Current Revision

2.2 The following document forms a part of this specification to the extent specified herein:

NASA Document MSFC-PROC-158B
Dated 15 February 1963
Procedure for Soldering of Electrical Connectors

3. REQUIREMENTS

3.1 Definition of Spacecraft Subsystems

The major and miscellaneous minor control items have been grouped together into functional groups as subsystems. These subsystems and the control items of which they are composed are listed below and shown diagrammatically in Figure 1.

1) Communication subsystem — 475025, 475030, 475040

2) Antenna control subsystem — 475035, 475303, 475307, 475302

3) Telemetry and command subsystem — 475045, 475050, 475055

4) Power supply subsystem — 475060, 250204

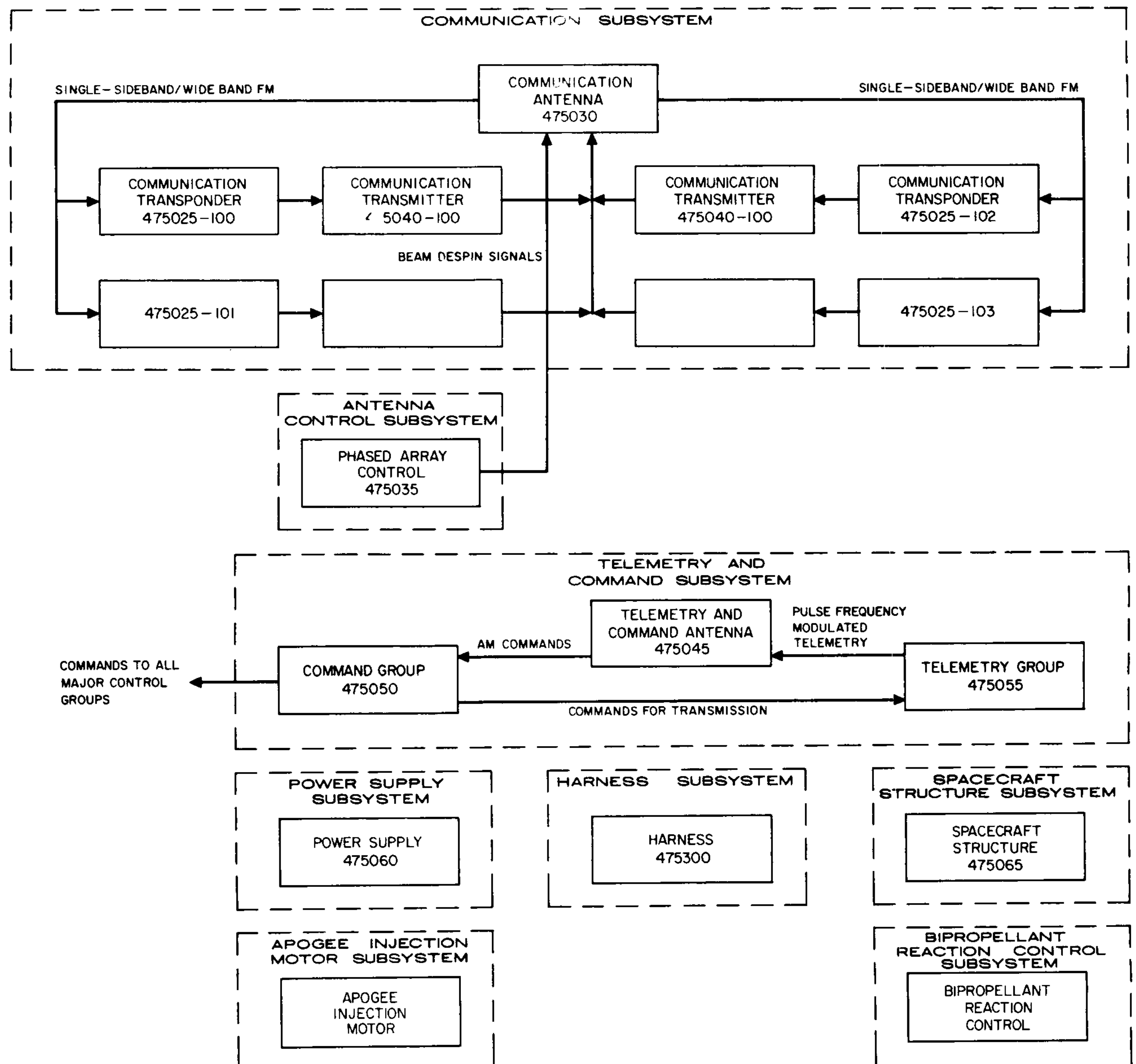

Figure 1. Subsystem and Major Control Items

5) Spacecraft structure subsystem — 475065, 475301, 475306, separation switch

6) Wire harness subsystem — 475300

7) Apogee injection motor subsystem

8) Reaction control subsystem — 250205

3.2 Communication Subsystem

The communication subsystem shall provide facilities to receive, convert frequency, amplify, and retransmit microwave signals. The communication subsystem shall be composed of the following major control items:

1) Communication transponders — 475025

2) Communication antenna — 475030

3) Communication transmitters — 475040

3.2.1 Reliability. The communication subsystem shall have a probability of operation within the performance requirements of 0.906 for 1 year and 0.525 for 3 years.

3.2.2 Modes. Each transponder shall be capable of operating in an FM to FM wideband mode (frequency translation) and a SSB to PM (multiple access) mode.

3.2.3 Beacon. An RF signal for ground station tracking shall be provided in both modes. In the multiple access mode the carrier shall be used. In the frequency translation mode, a separate beacon signal shall be provided.

3.2.3.1 Frequency Translation Beacon: The beacon frequency shall be derived from the frequency translation master oscillator. The frequency translation beacon frequencies shall be as given in Table 1.

3.2.4 Test Access Points. Test access points shall be provided on the receiving antenna output, preamplifiers, and traveling-wave tube outputs.

3.2.5 Telemetry Monitoring Provisions. A 0 to -5 volt dc output proportional to the received and transmitted signal strength shall be provided to the telemetry encoder.

3.2.6 RF Input Power. The nominal RF input power into the receiving antenna shall be -95 dbm for operation in the frequency translation mode and -92 dbm per channel test tone for operation in the multiple access mode.

TABLE 1. FREQUENCY ASSIGNMENTS (MEGACYCLES)

Transponder	Input*	Output**	Frequency Translation Beacon	Master Oscillator Multiple Access	Master Oscillator Frequency Translation
475025-100	6019.356	3992.112	4006.975	31.1882	15.83776
475025-101	6108.312	4051.108	4066.192	31.6491	16.01718
475025-102	6212.094	4119.938	4135.277	32.1870	16.34496
475025-103	6301.050	4178.934	4194.491	32.6979	16.57901

* Signal bandwidth = 25 mc in frequency translation mode
5.3 mc in multiple access mode

** Signal bandwidth = 25 mc

3.2.7 <u>RF Output Power.</u> The nominal RF output power of the transmitting antenna shall be 1.8 watts.

3.2.8 <u>Frequencies</u>. The communication subsystem shall operate at the assigned frequencies as shown on Table 1.

3.2.9 <u>Bandwidth</u>. The 1-db bandwidth for the frequency translation mode shall be 25 mc ± 1 mc measured between IF input and RF output. The 1-db bandwidth of the multiple access mode shall be 5.5 mc (+0.5mc, -0.2 mc) measured at the preamplifier output and 25 mc measured at the output of the transponder.

3.2.10 <u>Voltage Standing-Wave Ratio</u>. The voltage standing-wave ratio at the receiver and transmitter terminals, with all necessary cabling, multiplexers, and antennas properly installed on the spacecraft, shall not exceed 1.5:1 at the transmitting frequency and 2.5:1 at the receiving frequency.

3.2.11 <u>RF Impedance</u>. The input and output impedance of each microwave circuit element shall be nominally 50 ohms.

3.2.12 <u>Communication Transponder</u> - 475025

3.2.12.1 <u>Quantity</u>: There shall be four communication transponders.

3.2.12.2 <u>Common Mode Requirements</u>: The transponder, when operating in either the frequency translation or multiple access mode, shall meet the following requirements.

3.2.12.2.1 Noise Figure: The noise figure of each transponder receiver shall be less than 9 db (referenced to the standard noise temperature of 290° K).

3.2.12.2.2 RF Output Power: Each transponder mode shall have nominal power out of 1.7 milliwatts measured at the unit connector.

3.2.12.2.3 DC Power Requirements: Each transponder mode regulator shall require no more than 95 milliamperes from the -28-volt unregulated bus.

3.2.12.3 Frequency Translation Requirements: The frequency translation mode shall translate and amplify input signal frequencies with no conversion in modulation.

3.2.12.3.1 Received Carrier/Received Noise: The requirements of this specification shall not be imposed unless the received carrier power to the received noise power ratio is at least +17.2 db.

3.2.12.3.2 Long-Term Output Frequency Stability: The output frequency shall be stable to within 84 kc.

3.2.12.3.3 Delay Distortion: The delay distortion shall be less than

3.2.12.4 Multiple-Access Requirements: The multiple access mode shall convert the received single-sideband signals into a phase-modulated signal. This signal shall be multiplied to the proper microwave frequency and amplified.

3.2.12.4.1 Channel Capacity: Each of the multiple access modes shall be capable of conveying up to 1200 one-way 4-kc channels comprised of CCITT mastergroups 1, 2, 3 and 4.

3.2.12.4.2 Distortion Noise: Based on a compandor improvement factor of 15 db, the multiple access mode shall contribute less than 207,000 picowatts of distortion noise and 155,000 picowatts of intermodulation distortion noise. The phase delay distortion shall be less than

3.2.12.4.3 Short-Term Output Frequency Stability: The output frequency shall not vary at a rate greater than 5.46 cps/sec.

3.2.12.4.4 Long-Term Output Frequency Stability: The output frequency shall be stable to within 84 kc.

3.2.12.4.5 Test Tone/Intermodulation Noise Ratio: The test tone/intermodulation noise ratio shall be greater than 40.0 db for equivalent full channel loading.

3.2.12.4.6 Test Tone/Fluctuation Noise Ratio: The test tone/fluctuation noise ratio shall be greater than 47.7 db.

3.2.13 Communication Antenna – 475030. The antenna shall receive the incoming 6-gc signals, separate them into the four frequency channels, and supply them to the appropriate receiver. The antenna combines the four 4-gc signals from the transmitter, processes and transmits them.

3.2.13.1 Receiving Antenna: The receiving antenna shall be designed for reception of signals between 6.006 gc and 6.314 gc.

3.2.13.1.1 Pattern: The radiation pattern of the receiving antenna shall be omnidirectional within 0.5 db in the ϕ plane (perpendicular to the spin axis) and have a minimum 3-db beamwidth of 17.3 degrees in all planes (parallel to the spin axis). The peak of the beam will be at an angle of $\theta = 90$ degrees in all directions of ϕ.

3.2.13.1.2 Gain: The antenna pattern gain shall be at least 7.8 db over the input frequency range as given in Table 1.

3.2.13.2 Receiver Multiplexer: A multiplexer shall be provided to separate the received 6-gc signals into the four frequency channels.

3.2.13.2.1 Isolation: Isolation between frequency channels shall be at least 17 db at $f_o \pm 44.8$ mc (f_o denotes each of the four input frequencies shown in Table 1).

3.2.13.3 Loss: The total RF loss from the receiving antenna input to the transponder input shall be less than 3.0 db.

3.2.13.4 Phase Shifter Power Splitter: The RF power splitter shall split the RF power into eight equal amplitude and equal phase parts.

3.2.13.5 Phase Shifter: The phase shifters shall be capable of inducing in the communication RF signals (output frequencies as given in Table 1) 16 different phase shifts (ϕ_n) of $\phi_n = 2\pi \cos(\omega t + n \cdot 22.5 \text{ degrees})$, $n = 0, 1, 2, \ldots 15$, where the spacecraft spin rate, ω, shall be $100\pi \pm 100\pi$ radians.

3.2.13.5.1 Drive Characteristics

3.2.13.5.1.1 Drive Signals: The coil drive signals shall be as specified in paragraph 3.3.4.1.3.

3.2.13.5.1.2 Coil Input Impedance: The coil input impedance shall be nominally 125 ohms and 0.5 henry.

3.2.13.6 Transmitter Multiplexer: A multiplexer shall be used to combine the four 4-gc output signals.

3.2.13.6.1 Isolation: Isolation between channels shall be at least 17 db at $f_o \pm 44.8$ mc. (f_o denotes each of the four output frequencies listed in Table 1.)

3.2.13.7 Phased Array Transmitting Antenna: The phased array is defined as consisting of the 16 collinear arrays that make up the radiating portion of the antenna, the transmission lines leading from the outputs of the phase shifters to the element arrays, and any matching networks required to obtain a broad-band impedance match of the element arrays.

3.2.13.7.1 Gain: The pattern gain at the peak of the beam shall be at least 18.0 db for the input signals as given in paragraph 3.2.13.5.

3.2.13.7.2 Pattern Characteristics: The radiation pattern of the transmitting antenna for input signals as specified on paragraph 3.2.13.5 shall be an elliptically shaped pencil beam a minimum of 17.3 degrees wide in the θ plane (parallel to the spin axis) and 23 degrees wide in the ϕ plane (perpendicular to the spin axis). The peak of the beam will be at an angle $\theta = 90$ degrees for any direction of the beam in the plane.

3.2.13.8 Loss: The total RF loss from the transmitter output to the transmitting antenna output shall not exceed 3.5 db.

3.2.14 Communication Transmitter — 475040. The communication transmitter shall consist of a traveling-wave tube (TWT), TWT power supply, and input-output equipment.

3.2.14.1 Quantity: There shall be four communication transmitters. Each communication transmitter shall consist of two traveling-wave tubes and power supplies.

3.2.14.2 TWT and Power Supply: Each TWT shall be capable of amplifying output from either the multiple access mode or frequency translation mode of one transponder. The TWT power supply shall regulate voltage and provide, upon ground command necessary power for TWT operation.

3.2.14.2.1 RF Performance: Traveling-wave tube RF performance shall be as given in Table 2.

3.2.14.2.2 Voltage Input: The voltage input shall be nominally -31-volts unregulated.

3.2.14.2.3 Filament Turn On/Turn Off: A command pulse of +24 volts, 50 milliseconds, 2 milliamperes maximum shall turn on or off the filament voltage.

TABLE 2. TRAVELING-WAVE TUBE RF PERFORMANCE
TWT 384H

Minimum RF power output	4.0 watts for all inputs from 0.4 to 0.7 milliwatts
Maximum RF input	2.0 milliwatts
RF saturation gain	40 db
RF small signal gain	50 db
Noise figure	35 db maximum
Voltage phase modulation index	4 deg/volt maximum
RF drive phase modulation index	7 deg/db maximum

3.2.14.2.4 High Voltage Start: There shall be a nominal high voltage start pulse of +24 volts, 50 milliseconds, and 2 milliamperes maximum.

3.2.14.3 Loss: The total loss through the output equipment and cabling shall be less than 0.5 db. The total loss through the input equipment and cabling shall be less than 1.3 db.

3.3 Antenna Control Subsystem

The antenna control subsystem shall be comprised of the following items:

1) Phased array control — 475035

2) Sun sensor assembly — 475302

3) Central timer — 475303

4) Solenoid driver — 475307

This subsystem shall be capable of firing the apogee motor at the proper time, developing control signals to be used by on-board electronics to properly orient the spacecraft and keep it in the equatorial plane, maintaining antenna beam despin rate, and determining the angle between the spin axis and the spacecraft sun-line.

3.3.1 Reliability. The antenna control subsystem shall have a probability of operation within the performance requirements of 0.988 for 1 year and 0.963 for 3 years.

3.3.2 Telemetry Monitoring Provisions. Provision shall be incorporated for monitoring phased array control electronics (PACE) lock, central-timer selection, antenna beam position, and output of the ψ_2 angle counter for transmission to the ground via telemetry.

3.3.3 Test Access Points. Test access points shall be provided at the 2.81-minute output of the central timer, the PACE voltage-controlled oscillator output, and the output of the PACE pulse error gate.

3.3.4 Phased Array Control – 475035. The phased array control shall consist of:

1) Four phased array control electronics

2) Two phase shifter drivers

3.3.4.1 Phased Array Control Electronics (PACE): The PACE shall provide to the phase shifter drivers, control signals proportional to the spin rate to despin the communication RF signals. The PACE shall process signals supplied per paragraph 3.3.5 to measure, within 1 degree, the angle between the spin axis and the spacecraft sunline when the angle is within 90 ± 25 degrees.

3.3.4.1.1 Independence: Each PACE shall be able to operate independently of the others, but a maximum of one PACE shall be operative at any given time.

3.3.4.1.2 Input Signals: The PACE shall receive pulses from the ψ and ψ_2 sensors. (Characteristics of ψ and ψ_2 pulses are specified in paragraph 3.3.5.3.) During eclipse, these pulses shall be sent by ground command via the command subsystem. (Characteristic of command subsystem outputs shall be as specified in paragraph 3.4.7.2.8.) The PACE shall be capable of receiving a series of ground command pulses via the command subsystem to be used to place the beam at any angle ± 0.7 degree to the earth-spacecraft line. Characteristics of beam positioning pulses are specified in paragraph 3.4.7.2.8.

Time-of-day correction pulses (2.81 minutes/pulse) shall be received from all central timers to correct beam position for earth-sun relative motion. The PACE shall, upon ground command, select for use the pulses of one of the central timers. Characteristics of these pulses are specified in paragraph 3.3.6.2.

3.3.4.1.3 Control Signals: The PACE shall provide 16 control signals to the phase shifter drivers. They shall be of the form $\cos 2\pi \sin (2\pi ft + m\pi/8)$ and $\sin 2\pi \sin (2\pi ft + m\pi/8)$ where $m = 0, 1, 2, \ldots 7$.

3.3.4.1.3.1 Error: The control signal positioning error of the beam shall not lead or lag by more than 1 degree.

3.3.4.2 Jet Control Electronics — 475206: The jet control electronics (JCE) shall control the actuation of the axial and radial jet solenoids.

3.3.4.2.1 Modes: The jet control electronics shall operate in either a backup or an on-board mode.

3.3.4.2.1.1 On-Board Mode: The jet control electronics shall be capable, upon ground command, of firing the radial and axial jet solenoids during one-eighth of a spacecraft revolution with nominal pulse center at +90 degrees or -90 degrees from the antenna beam center (ϕ). (Characteristics of command subsystem outputs shall be as specified in paragraph 3.4.7.2.8.)

3.3.4.2.1.1.1 Position Sensing: The jet control electronics shall be capable of sensing the position of the axial and radial jets relative to the directional communication beam position.

3.3.4.2.1.2 Backup Mode: The jet control electronics shall be capable, upon ground command, of firing the axial and radial jet solenoids through any number of degrees of spacecraft rotation. (Characteristics of command subsystem outputs shall be as specified in paragraph 3.4.7.2.8.)

3.3.4.2.2 Output Signal: The jet control electronics shall provide an appropriate output to the solenoid driver.

3.3.4.3 DC Power Requirements: Each PACE and JCE shall have a common regulator and shall require no more than 30 milliamperes at -28 volts unregulated.

3.3.4.3.1 Turn On/Turn Off: Each regulator shall be capable of being turned on and turned off by separate turn-on/turn-off pulses. The pulses shall be 2 milliamperes maximum at +24 volts for 50 milliseconds.

3.3.4.4 Phase Shifter Driver — 475170: The phase shifter drivers shall be capable of linearly amplifying signals of the form cos $2\pi[\sin(2\pi ft$ plus $m\pi/8)]$ and sin $2\pi[\sin(2\pi ft$ plus $m\pi/8)]$, received from the PACE.

3.3.4.4.1 Output Signal: The output signals shall have a voltage swing of ±10 volts and a current swing of ±80 milliamperes.

3.3.4.4.2 DC Current Requirement: The total dc current requirement shall be 620 milliamperes maximum from both unregulated -28 volt buses.

3.3.4.4.3 Turn On/Turn Off: The phase shifter driver shall be capable of being turned on/turned off by a command pulse of +24 volt at 2 milliamperes maximum for 50 milliseconds.

3.3.5 Sun Sensor Assembly — 475302. Each sun sensor assembly shall provide signals for determining the angle between the spin axis and the sunline as well as synchronization data on the spacecraft spin rate.

3.3.5.1 General Description: Each sun sensor assembly shall consist of two ψ sun sensors and two ψ_2 sun sensors. The inner ψ and inner ψ_2 are alignment references. The outputs of these sensors will be utilized by the onboard electronics. The outputs of the outer ψ and outer ψ_2 will be telemetered. The beam plane of each ψ_2 sensor shall be at an angle of 35 ±0.5 degree with respect to its reference ψ sensor.

3.3.5.2 Installation and Alignment: Four sun sensor assemblies shall be used on each spacecraft. The sun assemblies will be placed around the circumference of the spacecraft at 45, 135, 225, and 315 degrees ±0.1 degree, relative to the reference radial jet. The beam plane of the reference ψ sensor will be parallel to the spin axis within ±0.25 degree. The beam plane of the reference ψ sensor will be parallel to a radial line of the spacecraft within ±0.1 degree. The line formed by the intersection of the ψ and ψ_2 planes will be perpendicular to the spin axis within ± degree.

3.3.5.3 Output Signal Characteristics: A sensor will provide an output signal when the sunline and its beam plane coincide.

3.3.5.3.1 ψ Sensors

3.3.5.3.1.1 Output Voltage: The sensor output voltage shall be between the limits shown in Figure 2 when the sunline and its beam plane coincide.

3.3.5.3.1.2 3-db Beam Width: The sensor 3-db beam width shall be $\left[\frac{0.8 \text{ degree} \pm 12.5 \text{ percent}}{\sin \phi_s}\right]$ for ϕ_s values from 30 to 150 degrees.

3.3.5.3.1.3 Sensor Triggering Level: The sensor shall have an output of mv at an angle of $\left[\frac{\theta_T \pm \quad \text{percent}}{\sin \phi_s}\right]$ from the center of its beam plane for ϕ_s values from 30 to 150 degrees.

3.3.5.3.2 ψ_2 Sensors

3.3.5.3.2.1 Output Voltage: The sensor output voltage shall be between the limits shown in Figure 2 when the sunline and its beam plane coincide.

3.3.5.3.2.2 3-db Beam Width: The sensor 3-db beam width shall be for ϕ_s values from 35 to 145 degrees.

3.3.5.3.2.3 Sensor Triggering Level: The sensor shall have an output of mv at an angle of $\left[\frac{\theta_T \pm \quad \text{percent}}{\sin \phi_s}\right]$ from the center of its beam plane for ϕ_s values from 35 to 145 degrees.

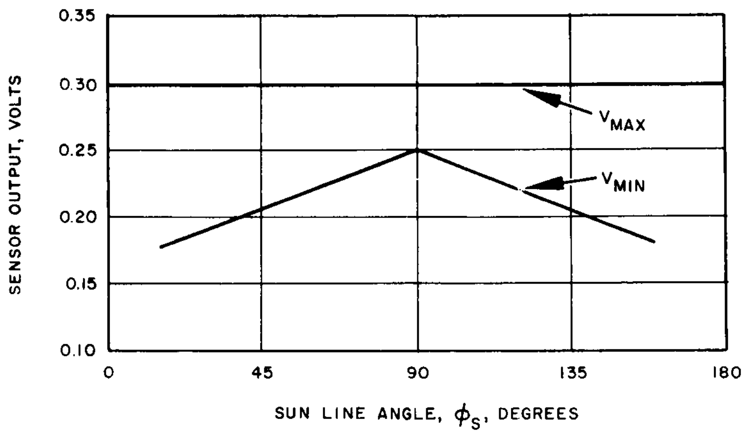

Figure 2. Sensor Output Voltage Limits

3.3.6 Central Timer — 475303

3.3.6.1 Quantity: There shall be four central timers.

3.3.6.2 Output Signals: The outputs shall be 24 ±0.7 volts, 4 milliamperes maximum for 20 ±10 milliseconds. Quiescent levels shall be 0.0 volt +0.1, -0.0 volt. The frequency (period) of the outputs shall be 315 minutes (+0, -21 seconds per pulse), 2.81 minutes per pulse, 194.18 cps. The tolerance on the frequencies (periods) shall be 0.02 percent.

3.3.6.3 Current Requirement: Each central timer shall require no more than 15 milliamperes at ±24 volts from its respective command regulator.

3.3.7 Solenoid Drivers — 475307. The solenoid drivers shall drive the fuel and oxidizer control solenoids.

3.3.7.1 Quantity: There shall be four solenoid drivers.

3.3.7.2 Input Signals: The input signals shall be from the jet control electronics.

3.3.7.3 Output Signal: The output signal shall be at least 20 volts at 0.9 ampere for 30 milliseconds for solenoid pull-in and 0.2 ampere at 4 volts for solenoid holding.

3.3.7.4 Impedance: The solenoid driver shall work into a nominal impedance of 75 millihenries and 20 ohms.

3.3.7.5 Current Requirements: The solenoid driver shall not require more than 1.5 amperes pull-in and 0.3 ampere hold from the unregulated -28 volt bus.

3.4 Telemetry and Command Subsystem

The telemetry and command subsystem shall provide facilities to receive, process, and execute commands to control spacecraft operation. It shall also provide facilities to encode digital and analog signals, which indicate quality of spacecraft operation, and transmit these signals to the ground.

The telemetry and command subsystem shall be comprised of the following items:

1) Telemetry and command antenna — 475045

2) Command group — 475050

3) Telemetry group — 475055

3.4.1 Reliability. The telemetry and command subsystem shall have a probability of operation within the performance requirements of 0.99 for 1 year and 0.98 for 3 years.

3.4.2 RF Power. The telemetry power transmitted shall be nominally 0.75 watt. The received command signal power shall be nominally -100 dbm.

3.4.3 Loss. The RF loss from the output of telemetry transmitters to the output of the telemetry and command antenna shall be less than 2.1 db.

3.4.4 Telemetry Monitoring Provisions. 0 to -5 volt dc outputs proportional to the telemetry transmitted power, contents of the command register, command encoder operating mode, and command enable power shall be provided.

3.4.5 Test Access Points. Test access points shall be provided on the whip antenna output and the command receiver output.

3.4.6 Telemetry and Command Antenna – 475045. The telemetry and command antenna shall consist of:

1) Two antenna groups

2) Eight whip antennas per group

3) Two diplexers per group

4) Two hybrid baluns

3.4.6.1 Whip Antenna Group

3.4.6.1.1 Polarization: The polarization of the radiation shall be elliptical. For transmission, the axial ratio of the polarization along the spin axis shall not be greater than 1 db.

3.4.6.1.2 Radiation Pattern: As nearly isotropic a coverage as possible shall be provided.

3.4.6.1.3 Bandwidth: The gain difference between receiving and transmitting frequencies shall not exceed 3 db.

3.4.6.2 Diplexer: One channel shall serve as an input to the receiver, and the other channel shall serve as an output from the transmitter.

3.4.6.2.1 Quantity: One diplexer for each telemetry/command group shall be provided.

3.4.6.2.2 Frequency: The input frequency shall be 148.260 mc (command) and the output frequencies 136.980 mc or 136.170 mc (telemetry).

3.4.6.2.3 Isolation: The receiver channel shall offer at least 68 db rejection to the telemetry transmitter frequency. The transmitter channel shall offer at least 58 db rejection to the receiver frequency.

3.4.7 Command Group – 475050. The command group shall be composed of:

1) Four command receivers – 475210

2) Four command filter-decoders - 475211

3) Four command regulators – 475212

3.4.7.1 Command Receiver – 475210: The receiver shall be designed to receive amplitude-modulated signals.

3.4.7.1.1 Frequency Band: The center of the frequency pass band shall be 148.260 mc.

3.4.7.1.2 Noise Figure: The noise figure of the receiver shall be 6 db maximum referenced to the standard source temperature of 290°K.

3.4.7.1.3 Receiver Stability: The receiver shall remain stable within 0.003 percent of the command frequency.

3.4.7.1.4 Image Response: The image frequency response level shall be at least 70 db below the response at the desired frequency as measured through the diplexer.

3.4.7.1.5 Audio Output: The nominal audio output shall be 2.2 volts rms.

3.4.7.1.6 Audio Output Response: The output frequency response shall be flat within 3 db from 1 dc to 12 dc.

3.4.7.1.7 Input Impedance: The RF input impedance shall be 50 ohms.

3.4.7.2 Command Decoder – 475211: The command decoder shall process the audio output signal of the command receiver so as to generate command signals to all required circuitry.

3.4.7.2.1 Redundancy: The command system shall be interconnected so that failure of one of the multiple systems does not compromise the ability of remaining systems to perform all command functions.

3.4.7.2.2 Standby Status: The command receivers and that portion of the decoder circuitry required to initiate full command turn-on shall be operating at all times that spacecraft power is on.

3.4.7.2.3 Modes: The command decoder shall operate in a shift mode (shifting bits received serially into the command register) or in a count mode (counting the number of received bits).

3.4.7.2.3.1 Shift Mode: The shift mode shall be the primary mode of the decoder. The shift mode shall operate from a non-return-to-zero (NRZ) format, frequency-shift-keyed (FSK) digital wave train. The bit sync shall be a 128 cps sine wave whose alternate zero crossing shall be at the time center of the data bits.

3.4.7.2.3.2 Count Mode: A 0 or 1 tone of 7.8125 milliseconds duration shall define a bit. An interruption of 12 bit times shall be required between bits.

3.4.7.2.4 Frequencies: The data bit frequencies shall be 7400 cps for a 0 bit and 8600 cps for a 1 bit. The execute frequency shall be 3620 cps.

3.4.7.2.5 Bandwidth: The bandwidth of the three tones specified in paragraph 3.4.7.2.4 shall be nominally 300 cps.

3.4.7.2.6 Format: The format for the Advanced Syncom commands shall consist of three major subdivisions: 1) word sync, 2) command address, and 3) command number. The address and command shall each be a 6-bit word.

3.4.7.2.6.1 Word Sync: The word sync shall be used to define the start time for the command which follows. For operation in the shift mode, word sync shall be a series of 25 zeros followed by a 1 and a 0. The word sync for operation in the count mode shall be 25 bits followed by a 12-bit time interrupt.

3.4.7.2.6.2 Command Address: After the word sync, the next 6 bits of the command message shall consist of a spacecraft number and a quadrant number to designate a specific command decoder. The format shall be as follows:

6	5	4	3	2	1
Most Significant Bit					Least Significant Bit

where bits 6, 5, 4, 3 are the spacecraft number and bits 2, 1 are the quadrant number.

3.4.7.2.6.2.1 Command Address Designations: The quadrants in each spacecraft shall be addressed as shown in Table 3.

TABLE 3. COMMAND ADDRESS WORD

Spacecraft Number/ Quadrant Number	Bit Number					
	6	5	4	3	2	1
T2-1	0	0	0	0	0	1
2	0	0	0	0	1	0
3	0	0	0	1	1	1
4	0	0	0	1	0	0
Y1-1 and Y2*	0	0	1	1	0	1
2	0	0	1	1	1	0
3	0	0	1	0	1	1
4	0	0	1	0	0	0
F1-1	0	1	1	0	0	1
2	0	1	1	0	1	0
3	0	1	0	0	1	1
4	0	1	0	0	0	0
F2-1	1	1	0	0	0	1
2	1	1	0	0	1	0
3	1	0	0	0	1	1
4	1	0	0	0	0	0
F3-1	1	0	0	1	0	1
2	1	0	0	1	1	0
3	1	0	1	1	1	1
4	1	0	1	1	0	0
F4-1	1	0	1	0	0	1
2	1	0	1	0	1	0
3	1	1	1	0	1	1
4	1	1	1	0	0	0

* Only one quadrant contained in Y2.

3.4.7.2.6.3 Command Number: The last group of 6 bits of the command message shall consist of the command number. The format shall be defined as follows:

6	5	4	3	2	1

Most Significant Bit — Least Significant Bit

Whenever applicable, bits 6, 5, 4, 3 shall designate the command type, and bits 2, 1 shall designate the appropriate quadrant.

A modified octal representation of the command shall be a three-digit code defined as follows:

1) The first digit shall be a 0 or 1 identical to the most significant binary bit in the register.

2) The second digit shall be a digit from 0 to 7 representing the next three most significant straight-binary-coded-decomal bits in the register.

3) The third digit shall be a digit from 1 to 4 representing the least significant straight-binary-coded-decimal bits the register. These bits shall define the quadrant, where applicable, quadrant 4 being represented by binary 00.

Examples:

Modified Octal to Binary

0	5	2	(Quadrant 2 TWT Filament on)
0	101	10	

Binary to Modified Octal

1	110	01	
1	6	1	(Fire axial jet system 1)

3.4.7.2.6.3.1 Command Designation: The command words shall be designated as shown in Table 4.

TABLE 4. COMMAND NUMBER WORD

	Command Function	Modified Octal Command	Bit Number 6	5	4	3	2	1
	Spares	00-1	0	0	0	0	0	1
		00-2	0	0	0	0	1	0
		00-3	0	0	0	0	1	1
		00-4	0	0	0	0	0	0
Transponder	Frequency translation mode on, multiple access mode off.	01-1	0	0	0	1	0	1
		01-2	0	0	0	1	1	0
		01-3	0	0	0	1	1	1
		01-4	0	0	0	1	0	0
	Multiple access mode on, frequency translation mode off	02-1	0	0	1	0	0	1
		02-2	0	0	1	0	1	0
		02-3	0	0	1	0	1	1
		02-4	0	0	1	0	0	0
	Spares	03-1	0	0	1	1	0	1
		03-2	0	0	1	1	1	0
		03-3	0	0	1	1	1	1
		03-4	0	0	1	1	0	0
	Transponder off	04-1	0	1	0	0	0	1
		04-2	0	1	0	0	1	0
		04-3	0	1	0	0	1	1
		04-4	0	1	0	0	0	0
Transmitter	TWT 1 filament on, TWT 2 filament off	05-1	0	1	0	1	0	1
		05-2	0	1	0	1	1	0
		05-3	0	1	0	1	1	1
		05-4	0	1	0	1	0	0

TABLE 4. (Continued)

	Command Function	Modified Octal Command	6	5	4	3	2	1
			Bit Number					
	TWT 2 filament on, TWT 1 filament off	06-1	0	1	1	0	0	1
		06-2	0	1	1	0	1	0
		06-3	0	1	1	0	1	1
		06-4	0	1	1	0	0	0
	TWT high voltage on	07-1	0	1	1	1	0	1
		07-2	0	1	1	1	1	0
		07-3	0	1	1	1	1	1
		07-4	0	1	1	1	0	0
PACE	PACE on for quadrant specified, all other PACE off	10-1	1	0	0	0	0	1
		10-2	1	0	0	0	1	0
		10-3	1	0	0	0	1	1
		10-4	1	0	0	0	0	0
	All PACE off	11-1	1	0	0	1	0	1
	Accept antenna position	11-2	1	0	0	1	1	0
	Power amplifier regulator on	11-3	1	0	0	1	1	1
	Power amplifier regulator off	11-4	1	0	0	1	0	0
	Eclipse mode operation on	12-1	1	0	1	0	0	1
	Eclipse mode operation off	12-2	1	0	1	0	1	0
	Timer select	12-3	1	0	1	0	1	1
		12-4	1	0	1	0	0	0

TABLE 4. (Continued)

	Command Function	Modified Octal Command	6	5	4	3	2	1
			Bit Number					
Telemetry	Telemetry on	13-1	1	0	1	1	0	1
		13-2	1	0	1	1	1	0
		13-3	1	0	1	1	1	1
		13-4	1	0	1	1	0	0
	Telemetry off	14-1	1	1	0	0	0	1
		14-2	1	1	0	0	1	0
		14-3	1	1	0	0	1	1
		14-4	1	1	0	0	0	0
Bipropellant system	Fire jets 1 and 2, +90 degrees axial	15-1	1	1	0	1	0	1
	Fire jets 1 and 2, +90 degrees radial	15-2	1	1	0	1	1	0
	Fire jets 1 and 2, -90 degrees radial	15-3	1	1	0	1	1	1
	Spare	15-4	1	1	0	1	0	0
	Fire axial jet, system 1 (backup mode)	16-1	1	1	1	0	0	1
	Fire radial jet, system 1 (backup mode)	16-2	1	1	1	0	1	0
	Fire axial jet, system 2 (backup mode)	16-3	1	1	1	0	1	1
	Fire radial jet, system 2 (backup mode)	16-4	1	1	1	0	0	0
Firing circuits	Fire control system squibs	17-1	1	1	1	1	0	1
	Release control system locking pins	17-2	1	1	1	1	1	0
	Fire apogee engine	17-3	1	1	1	1	1	1
	Spare	17-4	1	1	1	1	0	0

3.4.7.2.7 Operating Constraints

3.4.7.2.7.1 Execute: Execute shall take place in real time.

3.4.7.2.7.2 Inoperative: The execute channel shall be held inoperative by a continuous zero tone.

3.4.7.2.7.3 Clear: The command register shall clear every 315 minutes by an output from the local central timer. If a correct address is received without subsequent 0's transmission, the first 1's tone shall clear the register.

3.4.7.2.7.4 Bit Error: If a 1's tones is received during the 0's tone transmission normally applied, the decoder shall not turn off. The decoder shall turn off if the 0's tone transmission is interrupted for at least 10 milliseconds and followed by a 150 millisecond 0's or 1's tone transmission.

3.4.7.2.8 Output Signal: The command decoder shall provide for each received command an unloaded 24 volt, 2 milliamperes maximum pulse. The width of the command pulse shall be the length of the real time execute.

3.4.7.3 Command Regulator – 475212: The command regulator shall provide ± 24 volts ± 2 percent, -24 volts ± 3 percent to the local central timer, command receiver, and command decoder.

3.4.7.3.1 Input Power: The command regulator shall require a maximum of 40 milliamperes at -28 volts unregulated.

3.4.8 Telemetry Group – 475055. The telemetry group shall consist of:

1) Four telemetry transmitters – 475220

2) Four telemetry encoders – 475221

3) Four telemetry regulators – 475200

3.4.8.1 Telemetry Transmitter – 475220

3.4.8.1.1 Frequency: Two of the telemetry transmitters shall be designed to operate at a frequency of 136.470 mc and the other two transmitters shall be designed to operate at a frequency of 136.980 mc.

3.4.8.1.2 Frequency Stability: The transmitter frequency shall remain stable within 0.003 percent under normal service conditions.

3.4.8.1.3 Modulation Index: The transmitter frequency shall be phase-modulated, by the telemetry encoder output, with a 1.2 radian modulation index.

3.4.8.2 Telemetry Encoder – 475221: The telemetry encoder shall be capable of commutating the input signals and providing a suitable modulation signal to the telemetry transmitter.

3.4.8.2.1 Telemetry Format: The telemetry format shall be the GSFC standard PFM system, with approved exceptions. There shall be eight frames. Each frame shall be composed of 16 channels as follows:

1) Channel 0 shall be a frame sync nominal 5 millisecond reference burst followed by a nominal 15 millisecond sync burst.

2) Each channel of channels 1 to 7 shall contain a nominal 10 millisecond data burst preceded by a nominal 10 millisecond reference burst.

3) Channels 8 to 15 shall be a continuous data burst.

3.4.8.2.1.1 Bit Rate: The information bit rate shall be 48.5 ± 5 percent bits per second.

3.4.8.2.1.2 Clock: A nominal 194 cps input from the local quadrant central timer shall be used as a clock.

3.4.8.2.2 Encoder Output: The telemetry encoder shall convert digital and analog input signals to equal amplitude audio signals. The frequency of the audio tones shall be proportional to the input amplitude of the analog signals and to the digital number for digital signals. The frequency range shall be nominally 5 to 15 kc. The reference frequency shall be greater than 15.5 kc. The sync frequency shall be nominally 4.5 kc.

3.4.8.2.3 Special Telemetry: In addition to the format specified in paragraph 3.4.8.2.1, provision shall be made for telemetering the execute tone, and ψ and ψ_2 pulses.

3.4.8.2.4 Squib Firings: Provisions for indicating all squib firings shall be incorporated.

3.4.8.2.5 Solenoid Actuation: Provisions for indicating axial and radial jet solenoid actuation pulses shall be incorporated.

3.4.8.2.6 Channel Assignment: The telemetry channel assignment shall be as listed in Table 5.

TABLE 5. TELEMETRY CHANNEL ASSIGNMENT

Frame Number	Channel Number	Information
0	1	Spacecraft identification
0	2	Telemetry system identification
0	3, 4, 5, 6	Telemetry system calibration
0	7	Telemetry system power out
0	8-15	Transponder 1 received signal strength (continuous)
1	1	Bus current - unregulated bus 1
1	2	No. 1 battery voltage - unregulated bus 1
1	3	No. 2 battery voltage - unregulated bus 1
1	4	Bus current - unregulated bus 2
1	5	No. 1 battery voltage - unregulated bus 2
1	6	No. 2 battery voltage - unregulated bus 2
1	7	Spare
1	8-15	Transponder 2 received signal strength (continuous)
2	1, 2	Decoder 1 - enable, command register, mode
2	3, 4	Decoder 2 - enable, command register, mode
2	5, 6	Decoder 3 - enable, command register, mode
2	.7	Spare
2	8-15	Transponder 3 received signal strength (continuous)
3	1, 2	Decoder 4 - enable, command register, mode
3	3, 4, 5	ψ_2 angle
3	6	Bipropellant system No. 1 fuel pressure
3	7	Bipropellant system No. 1 fuel temperature
3	8-15	Transponder 4 received signal strength (continuous)
4	1	Bipropellant system No. 1 oxidizer pressure
4	2	Bipropellant system No. 1 oxidizer temperature
4	3	Bipropellant system No. 2 fuel pressure
4	4	Bipropellant system No. 2 fuel temperature

TABLE 5 (Continued)

Frame Number	Channel Number	Information
4	5	Bipropellant system No. 2 oxidizer pressure
4	6	Bipropellant system No. 2 oxidizer temperature
4	7	Spare
4	8-15	Bus voltage - unregulated bus 1 (continuous)
5	1, 2, 3	PACE - lock, timer selection, beam position
5	4	Transponder 1 - transmitter power
5	5	Transponder 2 - transmitter power
5	6	Transponder 3 - transmitter power
5	7	Transponder 4 - transmitter power
5	8-15	Bus voltage - unregulated bus No. 1 (continuous)
6	1	TWT 1 area temperature
6	2	TWT 2 area temperature
6	3	TWT 3 area temperature
6	4	TWT 4 area temperature
6	5-7	Spares
6	8-15	Bus voltage - unregulated bus 2 (continuous)
7	1	Solar panel temperature (different sensor for each encoder)
7	2-3	Structure temperature (different sensors for each encoder)
7	4-7	Spares
7	8-15	Bus voltage - unregulated bus 2 (continuous)

3.4.8.3 Telemetry Regulator — 475200: The telemetry regulator shall provide -24 volts, +33 volts ± 1 percent, and -33 volts ± 1 percent to the telemetry encoder and transmitter.

3.4.8.3.1 Turn On/Turn Off: The telemetry regulator shall be capable of being turned on and turned off by a standard command pulse from any of the command decoders. The characteristics of the command pulse are specified in Paragraph 3.4.7.2.8.

3.4.8.3.2 Input Voltage and Current: The input current to the regulator shall be 272 milliamperes maximum at -28 volts unregulated.

3.5 Power Supply Subsystem — 475060

The electrical power subsystem shall provide the spacecraft electronics with operating power. The primary source of electrical power will be solar panels. An electrical storage battery shall be provided to supply power during eclipses and peak transient loads.

3.5.1 Reliability. The power subsystem shall have a probability of operation within the performance requirements of 0.998 for 1 year and 0.994 for 3 years.

3.5.2 Current Distribution. The current shall be distributed to the system by two separate buses. The equipment on each bus shall be as shown in Table 6.

3.5.2.1 Current Distribution per Bus: Each bus shall provide power to two quadrants, one-half of the reaction control subsystem, phase shifter drivers, and one of the apogee motor squib firing circuits.

3.5.2.2 Bus Source: Each bus shall obtain its power from eight solar panels and two batteries.

3.5.3 Solar Array

3.5.3.1 Quantity: There shall be 16 solar panels.

3.5.3.2 Output: When initially placed in orbit the combined solar array output shall be a minimum of 27.2 volts and 5.14 amperes, at a solar intensity of 140 mw/cm^2, a solar cell temperature of 77°F, and a sun incidence angle of 90 degrees ± 25 degrees.

3.5.3.2.1 Ripple: The peak-to-peak ripple voltage output of the solar array, measured on the unregulated bus, shall not exceed 0.5 volt. The ripple frequency shall be less than 1000 cps.

TABLE 6. POWER DISTRIBUTION PER BUS

Item	Number of Units per Quadrant	Power per Unit, Milliamperes	Maximum Number of Operating Units per Bus	Bus 1 Units Operating	Bus 1 Load, Milliamperes	Bus 2 Units Operating	Bus 2 Load, Milliamperes
Telemetry transmitter	1	245	2	1	245		
Encoder	1	27	2	1	27		
Command receiver	1	27	2	2	54	2	54
Digital electronics (PACE)	1	30	1	1	30		
Phase shifters and phase shifter drivers	One unit common to both buses	620	1/2	1/2	310	1/2	310
Communication receivers	1	95	2	2	190	2	190
TWT (4-watt)	2	550	2	2	1100	2	1100
TOTAL					1956*		1654*

* Typical totals. Either bus must carry up to 2266 milliamperes.

3.5.4 Battery

3.5.4.1 Quantity: There shall be four batteries.

3.5.4.2 Battery Output: Each battery shall provide a minimum of 25.5 volts at the discharge depth of 20 percent.

3.5.4.3 Energy Capacity: The capacity shall be as follows:

6.0 amp-hr at 75°F at 1.2 amperes discharge rate

4.8 amp-hr at 100°F at 1.2 amperes discharge rate

4.8 amp-hr at 30°F at 1.2 amperes discharge rate

3.5.4.4 High Current Discharge: The cell terminal voltage during discharge of a fully charged cell at a load of 12.0 amperes shall be 1.0 volt minimum for a period of 10 seconds.

3.5.4.5 Discharge-Charge Efficiency: The discharge-charge efficiency is defined as the ratio of the ampere-hours removed from a fully charged battery during discharge to the ampere-hours required to restore it to its originally fully charged condition. For the initial system design, the discharge-charge efficiency shall be greater than 36 percent (including charge regulation).

3.5.4.6 Cycle Life: The minimum cycle life shall be 10,000 at 20 percent discharge depth.

3.5.5 Unregulated Bus Characteristics

3.5.5.1 Voltage: The unregulated bus voltage shall remain within the limits of -25.5 and -36 volts dc during normal operating conditions when in final stationary orbit with all equipment functioning properly. Necessary power regulation or conversion shall be provided as part of each subsystem.

3.5.5.2 Transients: During transfer of spacecraft equipment loads the voltage at the equipment terminals shall recover to within the steady-state limits in less than 0.5 second.

3.5.6 Undervoltage Protection. Undervoltage protection shall be provided by each regulator.

3.6 Structure Subsystem

3.6.1 Structure. The spacecraft structure shall provide the basic support for the other subsystems of the spacecraft and for attachment to the spacecraft support structure of the boost vehicle.

3.6.2 Reliability. The structure subsystem shall have a probability of operation within the performance requirements of 0.997 for 1 year and 0.991 for 3 years.

3.6.3 Nutation Damper — 475301. The spacecraft shall utilize two nutation dampers to dissipate the spacecraft nutation energy as heat. The time constant of this device shall be less than 1 hour.

3.6.4 Squib Driver Circuits. The squib driver circuits shall provide, upon command, an output pulse to fire the apogee injection motor ignition squibs, normally closed bipropellant valve squibs, and axial jet locking pin squibs.

3.6.4.1 Quantity: There shall be 12 squib driver circuits.

3.6.4.2 Reliability: The reliability of the squib driver circuit shall be at least 0.999.

3.6.4.3 Output Signal: The squib driver circuit shall provide an output of 15 ± 5 amperes for 10 milliseconds ± 5 milliseconds.

3.6.4.4 Output Impedance: The output impedance shall be 4 ± 2 ohms.

3.6.4.5 Input Trigger: The input trigger pulse shall be nominally 100 milliseconds from either the command decoder or central timer.

3.6.5 Separation Switch. Six separation switches shall be used to complete the circuit between the apogee motor squib and squib drivers and to remove the reset from the output stages of the central timers. The separation switch shall be actuated when the spacecraft separates from the Agena D vehicle.

3.7 Wire Harness Subsystem — 475300

3.7.1 Wire

3.7.1.1 Voltage Drop: The electrical power and signal distribution system shall be so designed that, under normal operating conditions, at no time shall any terminal voltage fall below the rated value due to excessive voltage drop in response to transmission of rated currents.

3.7.1.2 Mechanical Strength: No wires smaller than size AWG-26 or equivalent etched circuit lines shall be used in the electrical power and signal distribution system.

3.7.2 Harness Construction. The harness shall be constructed to minimize noise effects and to minimize physical damage from heat, flexing, and centrigugal force.

3.7.3 Twisting. Twisting shall be used whenever necessary to eliminate noise effects.

3.7.4 Shielded Wire. There shall be no electrical connections between the shield of any shielded wire and any electrical circuitry. The shield of a shielded wire shall be grounded only at one end.

3.7.5 Grounding

3.7.5.1 Common Ground: System return leads requiring grounding shall be terminated as close as possible to the positive battery ground terminal.

3.7.5.2 Unit Case Grounds: A component case may be considered a shield. One connector pin on each unit may be electrically connected to the unit case internally. The case may be grounded by mechanically contacting a system ground plane or by wiring to the case ground connector pin.

3.7.6 Soldering. All soldering operations to be performed on Advanced Syncom must be done by personnel that have been certified as passing requirements set forth in NASA Document MSFC-PROC-158B, "Procedure for Soldering of Electrical Connectors," (High Reliability) 15 February 1963.

The following numbers indicate paragraph numbers of the referenced specification to which exception shall be taken.

VI A 2c — If tubing cannot be used at abrasion points, wrapping may be substituted. If tubing cannot be fitted over a terminal, it may be dispensed with if the terminal has been correctly soldered, and sufficient clearance exists. MIL-I-22129 is waived.

VI A 1d — Preparation of new terminal should not be necessary. Terminals that are being reworked may be cleaned as necessary, care being taken not to damage the terminal or surrounding area.

VI A 1g — Silver-plated wire presently utilized on Syncom II is insulated with fluorinated ethylene propylen (FEP) and is acceptable.

VI A 2a — Any thermal stripper, such as American Missile Products MOD. WS-17B, Ideal Model No. 45-130 or 45-141, that, in the opinion of the Quality Control inspector, does a satisfactory job, is authorized.

VI A 2d — Heat sink tools will continue to be used. However, the circumferential edges of the tool will be rounded and burnished to reduce the possibility of wire abrasion.

VI A 2h — Resistance soldering will not be used until it has been thoroughly tested and established as a satisfactory technique.

VI A 3b — The environmental conditions described cannot at present be met. However, all good housekeeping procedures will be followed, as described in VI A 3a.

To reduce the possibility of accidental damage from loose tools, soldering technicians will use tool trays.

Harness boards will be cleaned regularly to avoid inclusions of debris in the harness assemblies.

VI A 3e(2) — The exposed wire between pot and insulation shall be limited to 0.1 inch. If by chance this tolerance is exceeded, NASA will consider and may approve the condition on an individual basis.

VI B 1d — The solder should follow the cup contour as closely as is reasonable under the particular circumstance and should have a slightly concave appearance. However, a convex configuration is acceptable provided the solder does not protrude beyond the outer diameter of the pot. It is recognized that the soldering process may leave a small amount of solder adhering to the outer surface of the pot, but excessive amounts, in the opinion of the Quality Control inspector, shall not be allowed. Inspections will be made with five to ten power magnifiers. In cases of controversy, a higher power magnification may be used.

VI B 2a — Where design or layout necessitates the connecting of more wires than the terminal was designed for, the extra wires may be wrapped around the terminal. All caution will be used to ensure proper clearance, insulation, and correct soldering. A sufficient excess length of wire may be used on terminal posts to permit one wiring change. These connections shall be heavily coated with protective paint to provide support as well as insulation.

VI B 1f — Wicking should not extend beyond 0.25 inch from the solder pot and connections which exceed this limit will be rejected if the insulation is bulged.

VI B 3 — Where necessary, in the opinion of the Quality Control inspector, the wrap may be increased up to 270°F.

3.7.7 Filtering. RF filters shall be provided on power line inputs to RF circuitry.

3.7.8 Terminal Boards. Terminal boards may be used, when a significant increase in reliability is provided.

3.8 Apogee Motor Subsystem

The apogee motor subsystem shall consist of a solid-propellant rocket motor to be used for injection of the Advanced Syncom spacecraft into a nominally circular equatorial orbit from the apogee of a transfer orbit. This subsystem is GFE. The requirements for this subsystem are defined in Procurement Specification for Syncom II Apogee Rocket Motor; Buyer: National Aeronautics and Space Administration; Contractor: Jet Propulsion Laboratory.

3.8.1 Ignition Squibs. The ignition squibs shall be fired by the squib driver circuits by either an on-board fixed timer or command from the ground.

3.9 Reaction Control Subsystem

The reaction control subsystem shall provide two redundant sources of thrust to correct spacecraft longitude, provide the spacecraft with the capability of being synchronous in the equatorial plane, maintain spacecraft spin rate, and orient the spacecraft so that the antennas will illuminate the earth continuously, within limits of proper performance by the apogee injection motor and the launch vehicle. The subsystem shall consist of storage tanks for fuel and oxidizer, injector solenoid valves, fuel and oxidizer lines, and thrust chambers aligned axially and radially. The requirements for this subsystem are defined in Hughes Specification X-254044, Procurement Specification for Advanced Syncom Bipropellant Reaction Control System.

3.9.1 Normally Closed Valve Squibs. The normally closed valve squib shall be fired by the squib driver circuit.

3.9.2 Locking Pin Squib. The locking pin squib shall be fired by the squib driver circuit.

3.9.3 Radial Jet Alignment. The radial thrust chamber shall be located so that its reaction force vector passes through the geometric centerline of the spacecraft with a tolerance of   inch.

The radial thrust chamber shall be located so that its reaction force vector passes within   inches of the longitudinal center of gravity of the spacecraft after apogee injection motor burnout.

3.9.4 Axial Jet Thrust Alignment. With the spacecraft spinning at 100 ± 5 rpm, the axial jet thrust chamber force vector shall lie in a plane which contains the geometric centerline and be parallel to the geometric center within ± 3 degrees.